practice is shown cross-hatched. Similarly, in FIG. **3**, a design range according to the present invention for the turbofan aircraft engine of FIG. **1** is shown unidirectionally hatched in a diagram of the total pressure ratio and a total stage count n_{St} of the second turbine. For comparison, a design range according to previous in-house practice is shown cross-hatched.

[0036] As indicated in FIG. **2** by a double-dot-dash line, the second turbine of the turbofan aircraft engine of FIG. **1** is designed such that, at a predetermined design point, especially at the "redline point" or a point of maximum allowable rotational speed and maximum allowable mass flow rate, a quotient of the total blade count N_{BV} of the second turbine divided by 100 is less than a difference of the total pressure ratio (p_1/p_2) of the second turbine minus one. In FIGS. **2** and **3**, a short-dashed line indicates that the total pressure ratio of the second turbine is greater than 4.5.

[0037] As indicated in FIG. **3** by a dot-dash line, the stage pressure ratio Π of one or more, in particular all, of the turbine stages of the second turbine is at least 1.6. As indicated in FIG. **3** by a long-dashed line, the total stage count n_{St} of the second turbine is at least two and no greater than five.

[0038] In FIG. **4**, a design range according to the present invention for the turbofan aircraft engine of FIG. **1** is shown unidirectionally hatched in a diagram of a product An^2 of an exit area A_L (see FIG. **1**) and a square n^2 of a rotational speed n and a blade tip velocity u_{TIP} of the second turbine. For comparison, a design range according to previous in-house practice is shown cross-hatched. Similarly, in FIG. **5**, a design range according to the present invention for the turbofan aircraft engine of FIG. **1** is shown unidirectionally hatched in a diagram of the product An^2 of the exit area and the square of the rotational speed and the stage pressure ratio Π of one or more, in particular all, of the turbine stages of the second turbine. For comparison, a design range according to previous in-house practice is shown cross-hatched.

[0039] As indicated in FIGS. **4**, **5** by a dashed line, the second turbine of the turbofan aircraft engine of FIG. **1** is designed such that a product An^2 of an exit area A_L of the second turbine and a square n^2 of a rotational speed n of the second turbine at the predetermined design point is at least 5.1010 [$in^2 \cdot rpm^2$] or 8961 [m^2/s^2], respectively. As indicated in FIG. **5** by a dot-dash line, the stage pressure ratio Π of one or more, in particular all, of the turbine stages of the second turbine is at least 1.6.

[0040] As indicated in FIG. **4** by a double-dash-dotted line, a blade tip velocity u_{TIP} of at least one turbine stage, particularly of the first or last turbine stage, of the second turbine at the predetermined design point is at least 400 meters per second.

[0041] Although the above is a description of exemplary embodiments, it should be noted that many modifications are possible. It should also be appreciated that the exemplary embodiments are only examples, and are not intended to limit scope, applicability, or configuration in any way. Rather, the foregoing description provides those skilled in the art with a convenient road map for implementing at least one exemplary embodiment, it being understood that various changes may be made in the function and arrangement of elements described without departing from the scope of protection set forth in the appended claims and their equivalent combinations of features.

LIST OF REFERENCE NUMERALS

[0042] A_B inlet area of the secondary duct
[0043] A_C inlet area of the primary duct
[0044] A_L exit area of the low-pressure turbine
[0045] B secondary duct (bypass)
[0046] BK combustion chamber
[0047] C primary duct (core)
[0048] D_F maximum blade diameter of the fan
[0049] D_L maximum blade diameter of the low-pressure turbine
[0050] F fan
[0051] G transmission (speed reduction mechanism)
[0052] HC (high-pressure) compressor
[0053] HT first turbine or high-pressure turbine
[0054] L second turbine or low-pressure turbine
[0055] W**1** hollow shaft
[0056] W**2** low-pressure shaft
[0057] p_1/p_2 total pressure ratio of the second turbine
[0058] N_{BV} total blade count of the second turbine
[0059] n_{St} total stage count of the second turbine
[0060] Π stage pressure ratio of the second turbine
[0061] An^2 product of the exit area A_L of the second turbine and the square of the rotational speed n
[0062] u_{TIP} blade tip velocity of the second turbine

What is claimed is:

1. A turbofan aircraft engine comprising:

a primary duct including a combustion chamber, a first turbine disposed downstream of the combustion chamber, a compressor disposed upstream of the combustion chamber and coupled to the first turbine, and a second turbine having a plurality of turbine stages having rotor blades and disposed downstream of the first turbine and coupled via a speed reduction mechanism to a fan for feeding a secondary duct of the turbofan aircraft engine;

the second turbine having a total stage count (n_{St}) of all turbine stages of the second turbine, a total blade count (N_{BV}) of all rotor blades and stator vanes of all turbine stages of the second turbine, a stage pressure ratio (Π) of the pressure at the inlet to the pressure at the outlet at each turbine stage, and a total pressure ratio (p_1/p_2) of the pressure at the inlet of a first turbine stage to the pressure at the exit of a last turbine stage of the second turbine at a design point,

a quotient ($N_{BV}/110$) of the total blade count divided by 110 being less than a difference ($[(p_1/p_2)-1]$) of the total pressure ratio minus one, with the total pressure ratio being greater than 4.5; and

at least one stage pressure ratio is at least 1.5; and

the second turbine having at least two and no more than five turbine stages; and/or

a quotient ($(p_1/p_2)/n_{St}$) of the total pressure ratio divided by the total stage count being greater than 1.6.

2. The turbofan aircraft engine as recited in claim **1** wherein each stage pressure ratio is at least 1.5.

3. The turbofan aircraft engine as recited in claim **1** wherein a quotient ($N_{BV}/100$) of the total blade count divided by 100 is less than the difference of the total pressure ratio minus one; and/or the total pressure ratio is greater than 5; and/or at least one stage pressure ratio is at least 1.6, in particular at least 1.65; and/or the turbine has no more than four turbine stages.

4. The turbofan aircraft engine as recited in claim **3** wherein each stage pressure ratio is at least 1.6.

5. The turbofan aircraft engine as recited in claim **3** wherein at least one stage pressure ratio is at least 1.65.